Paper 14 presented at the Amer~~ican~~ Astronautical Soc~~iety~~ Sympos~~ium~~ on Rocky Mountain Resources for Aerospace Scien~~ce~~ and Technol~~ogy~~, Denver, ~~Colorado~~, July 13-14, 1967. 20

ELECTRIC PROPULSION FOR SATELLITES*

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SUMMARY

Electric propulsion is achieving a competitive status for a number of satellite missions. Beyond the present status there are a number of advanced concepts that have promise of higher performance, and consequently a broader applicability to space propulsion. Most of these concepts have features which avoid or diminish fundamental limitations of existing thrusters, and in addition have important advantages in compatibility with the overall spacecraft system.

Auxiliary propulsion systems for attitude and position control of satellites and spacecraft must meet exacting requirements for thrust magnitude and direction, and must be compatible with the power supply. Advanced concepts with promise of high performance in meeting these requirements include electron-bombardment, radioisotope-heated contact-ion, plasma-separator, plasma, and colloid thrusters.

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* This review was carried out under NASA Grant NGR06-002-032, Electric Thruster Systems, OART.

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INTRODUCTION

Electric propulsion has received a considerable research and development effort in the past 10 years. This research and development effort has produced electric propulsion systems that are now ready for use in space missions (1-5). Beyond the present status there are a number of advanced concepts in electric propulsion that have promise of higher performance(6) and consequently a broader applicability to space propulsion. It is the purpose of this paper to briefly review these advanced concepts and to compare their potential performance with the performance of existing flight-prototype electric propulsion systems. A more detailed review is given elsewhere(7).

Most of these advanced concepts have features that avoid or diminish fundamental limitations of existing thrusters. For example, conventional electrostatic thrusters have a low thrust per unit area at lower values of specific impulse. This limitation on the performance of conventional electrostatic thrusters is due to the fundamental laws of space charge flow. Two of the advanced concepts have features which diminish the fundamental limitation of space charge. In one of the advanced concepts, charged particles are used which are much more massive than atomic ions thereby providing much greater thrust density. The other advanced concept employs very closely spaced electrodes, thereby greatly reducing the space charge and allowing operation at reasonably high thrust densities and low specific impulse.

Some of the advanced concepts have important advantages in compatibility with the overall spacecraft system. For example, two of the advanced concepts would be operated with the direct output from solar-cell arrays in contrast to existing flight-prototype ion thrusters which require complex power conditioning to change the direct current solar cell power into high-voltage d.c. power.

As a part of the comparison between the various thruster types and concepts it is desirable to denote their state of realism. Three general classifications are used in this paper:

1. "Flight-prototype" refers to extensively tested complete systems that are based on well-known technology throughout, and that are ready for starting on a flight program.
2. "Experimental" refers to electric thruster systems that are fairly well tested in laboratory-prototype form, with a number of man years of research and development, and in which all of the basic principles have been demonstrated.
3. "Advanced concept" is an electric propulsion system that is based on demonstrated principles, which has not been operated as a laboratory prototype.

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Classification of the various thruster systems into these three catagories is based on published information, and on past experience in the field of electric propulsion.

POWER FOR ELECTRIC PROPULSION

At the present time solar-photovoltaic cells are the favored source of power for electric propulsion. No serious difficulty is anticipated in using solar cells to power small electric thrusters for auxiliary propulsion(2). The solar intensity in space at Earth distance from the sun is about 1.4 kilowatt/sq. meter. Solar-cell efficiencies are less than 10%, so a solar array will deliver about 140 watts of electric power per sq. meter of array area. Some of the most demanding tasks for auxiliary propulsion is the north-south station keeping of synchronous satellites. An electric thruster operated continuously at a level of about 350 micro-pounds could perform the north-south station keeping function, and such a thruster requires about 100 watts of electric power. For this application of auxiliary electric propulsion, a solar array of about three feet by three feet would provide all the power needed. Such an application of solar cells is certainly within the state of the art.

Solar-cell arrays at power levels below 100 watts have specific weights of about 150 lb/kwe(8). This value of specific weight is used in this paper for comparing system weights of the various types of auxiliary electric propulsion.

Solar-cell arrays have several disadvantages in some mission applications(8). Solar-cell arrays in the 100 watt range and above must be packaged during boost, and unfolded after boost. The array must be sun-oriented, which introduces considerable mechanical complexity in most missions. In those missions where a substantial amount of time is spent in the planet shadow, batteries must be provided for energy storage, which results in very large increases in system weight(9).

The radioisotope thermoelectric generator is a potential power source for electric propulsion. Specific weight of the RTG used in the Transit satellites is about 1000 lb/kwe. Because of this very high specific weight, RTG's probably could be used only for auxiliary electric propulsion in the 10 watt range. If nuclear safety requirements can be relaxed in the future then the RTG may have a lower specific weight which would allow application to auxiliary propulsion at higher power levels. For those missions where a large fraction of time is spent in the planet shadow, it may well be that RTG's will prove superior to the solar-cell battery system. Another potential application for the RTG is that of auxiliary-propulsion power at great distances from the sun.

POWER-CONDITIONING AND CONTROLS

Most electrostatic thrusters and some plasma thrusters require high-voltage d.c. electric power. Typical solar-cell arrays have an output of 28 volts. d.c. Because of the mismatch in voltage between the powerplant and the electric thruster, power-conditioning is usually a necessary part of the electric propulsion system.

It will be shown later that power conditioning can be a major fraction of the total propulsion system weight. The power conditioning and controls weights shown in Figure 1 are representative of the present state-of-the-art for flight-prototype systems. The weight of 5 lbs at 27 watts of output power from the power conditioning is the total weight of the flight-prototype power conditioning for a 14 micro-pound electrostatic thruster(10). The weights shown in Figure 1 above 1000 watts are taken from a flight-prototype power conditioning and controls system designed for primary solar-electric propulsion(7). The controls weight is about 0.5 pounds, so the power conditioning accounts for nearly all of the weight. This is not too surprising because the power conditioning consists of a large number of components in a d.c./a.c. inverter, a transformer stage, and a a.c./d.c. rectifier stage.

Power losses in the power conditioning can result in very appreciable increases in system weight, because these losses increase the size and weight of the electric powerplant. The power conditioning and controls efficiency of 65% at 27 watts of output power is representative of values encountered in a 14 micro-pound electrostatic thruster system(10), but it is possible that this efficiency would be improved. The efficiencies of 92% and better above an output power of 1000 watts are taken from a flight-prototype design intended for primary solar-electric propulsion(7).

In addition to the substantial weight and power losses caused by the power conditioning and controls, this subsystem also introduces a very much greater complexity to the total electric propulsion system. For example, the power conditioning and controls for the 14 micro-pound electrostatic thruster has more than 1000 critical components. The ramifications to system reliability are clearly of major importance. It is self-evident that a truly optimum design will be achieved only when the electric thruster can operate directly from the powerplant output with a bare minimum of power conditioning, with very simple control circuitry, and low-weight power cables.

AUXILIARY ELECTRIC PROPULSION

Electric propulsion systems are well suited for a number of auxiliary propulsion functions which include:

- satellite station-keeping
- satellite attitude control
- satellite drag cancellation
- satellite orbit changes
- other programmed maneuvers
- mid-cource control of spacecraft attitude

Some of these thrusting functions, particularly in very long duration missions, are so demanding that it appears that electric propulsion will be the only satisfactory system.

An excellent example of the potential application of electric propulsion is the station-keeping and attitude-control of synchronous satellites(2). A typical electric propulsion system design for the synchronous satellite is shown in Figure 2. Approximate ΔV requirements for the synchronous satellite mission are:

north-south station-keeping	180 ft/sec/yr
east-west station-keeping	7 ft/sec/yr
attitude-control	13 ft/sec/yr

For a 2000-pound satellite these ΔV's correspond to a total impulse of about 12,000 lb-sec/yr. Propellant weight is simply the total impulse divided by specific impulse. For example, a propulsion system with a specific impulse of 100 seconds would require 120 pounds of propellant per year of mission time. If the synchronous satellite were intended to have a lifetime of 5 years, then one fourth of the total satellite weight would be propellant. Electric propulsion systems have specific impulses of the order of 1000 seconds so that the propellant requirement can be greatly reduced.

Electric power required by the thruster is proportional to the product of thrust and specific impulse. Because of the high specific weight of solar-cell arrays it is most advantageous to operate the electric thrusters at low thrust levels but for long thrusting times. In this way the power required by the thruster can be minimized, and the combination of low thrust and long thrusting time can still satisfy the total impulse requirements. For the continuous thrusting mode of operation shown in Figure 2, the thrust levels for a 2000-pound synchronous satellite are:

north-south station-keeping	350 micro-pounds
east-west station-keeping	14 micro-pounds

There are exceptions to the continuous-thrust mode of operation, for example a continuous flow of electric power from the solar-cell array could be stored in capacitors. When sufficient energy is accumulated in the capacitors the energy can be discharged to the electric thruster thereby producing repetitive impulse bits each having a thrust level much higher than the values listed above. In both of these modes of operation the power level of the solar-cell array is minimized, and in this way the weight of the total system is also kept to a minimum.

Flight-Prototype Electric Propulsion Systems

There are three types of electric propulsion systems that can be considered ready for use in a flight program. These systems are all powered by solar-cell arrays and can be distinguished by the type of thruster: resistojet, contact-ion electrostatic thruster, and electron-bombardment electrostatic thruster. All components of these systems are derived from well-known technology(11). Furthermore, each of the thruster types has been duration tested for many thousands of hours in ground vacuum facilities. As a final accreditation for the flight-prototype status, all three of the thruster types have been flown in space.

Resistojets. The present status of flight-prototype resistojet systems is described in detail in another paper(12), so the discussion here will be brief. In resistojets, electric power is used to resistance-heat surfaces over which the propellant flows. The heated propellant is then expanded thermodynamically through a conventional exhaust nozzle. In principle, a specific impulse of about 1000 seconds should be possible using hydrogen as the propellant.

Because of the high tank weights required for long-term storage of liquid hydrogen, liquid ammonia is used(9). The ammonia is vaporized and dissociated so that the propellant is a mixture of nitrogen and hydrogen gas. In the micro-pound thrust range the ammonia resistojet has achieved specific impulses of 175 to 250 seconds. These relatively low values of specific impulse appear to be due to the relatively moderate temperatures achieved so far, and the very small nozzle throat size which causes considerable loss due to slip flow.

A resistojet with a specific impulse of 200 seconds would require about 56 pounds of propellant to accomplish the north-south station keeping of a 2000-pound satellite for only one year. As will be shown in the following sections the weight of resistojet systems is much greater than flight-prototype ion thruster systems for performing such missions as the synchronous satellite. However the resistojet in the micro-pound range may find use in important functions such as the inversion maneuver for gravity-gradient stabilized satellites(2). Resistojets in the micro-pound thrust class have important advantages for moderate total-impulse missions; the electric power required is less than 10 watts, the resistance heater can be designed to operate directly from the solar-cell array output power, and the resistojet itself is very simple and rugged. Thrust vectoring probably would have to be accomplished by a mechanical gimbal.

Typical performance of flight-prototype resistojets is summarized in Table I. Propulsion times of 1000 and 10,000 hours are used here for purposes of comparison with the other types of auxiliary electric propulsion systems. As noted above the high weight of resistojet systems for long propulsion times is due to the low values of specific impulse that have been obtained with resistojets in the micro-pound thrust range. This poor performance for long propulsion times is in contrast to the superior performance of resistojets for those missions where propulsion times of the order of 100 hours are required.

Contact-ion thrusters. Research and development of the contact-ion thruster began as early as 1957, and since then a major effort has been expended to bring this thruster type to the flight-prototype stage. The contact-ion thruster has tens of thousands of hours of duration testing in ground vacuum facilities, and has been successfully flown in space.

In contact-ion thrusters, cesium propellant moves from the storage tank to a vaporizer and flow control by capillary flow. The cesium vapor at several torr pressure flows through the porous tungsten ionizer and is ionized as it leaves the ionizer. This contact ionization process proceeds with nearly 100% effectiveness when the ionizer is maintained at 1150 to 1350 deg K. The ionizer is maintained at a voltage above space potential and the accelerator electrode is maintained at a voltage below space potential to prevent electron backstreaming. The cesium ions are accelerated through the electrostatic field and emerge at high exhaust velocity from the ionizer. The ion exhaust beam must be neutralized with electrons which are supplied from the neutralizer filament. Very efficient heat shielding must be used to prevent excessive heat loss from the ionizer.

Mass flow rate in the ion exhaust beam is proportional to the ion current. Exhaust velocity of the ions is proportional to the square root of the net accelerating voltage as shown in Figure 3. Because of the space-charge in the acceleration region, the ion current density is limited to a maximum value which is directly proportional to the 3/2 - power of accelerator voltage and inversely proportional to the square of the spacing between the accelerator electrodes. For precision ion optics the accel length cannot be much less than the ionizer width. Fabrication difficulties have prevented accel lengths of less than a few millimeters, so that the accelerator voltages must be in the kilovolt range in order to provide a reasonably high value of thrust density(13). For this reason electrostatic ion thrusters require much power conditioning to convert the low voltage power from solar-cell arrays.

Electrostatic acceleration of ions in the contact-ion thruster provides a simple means for precision thrust vectoring. By applying potential differences to segments of the accelerator electrode, or to electrodes downstream of the accelerator electrode, the ion beam may be deflected in a manner similar to a cathode ray tube(14,15). A microthruster with segmented electrodes is shown in Figure 4. Precision thrust vectoring has been amply demonstrated throughout at 32 deg. cone angle.

Complete flight-prototype contact-ion microthruster systems(10,16) (except for solar-cell arrays) are shown in Figures 5 and 6. Thruster systems such as these are intended for the east-west station keeping and the attitude control functions of synchronous satellites illustrated in Figure 2. A contact-ion thruster (16) suitable for the north-south station keeping function is shown in Figure 7.

Electron-bombardment thrusters. Research and development on electron-bombardment thrusters started in about 1960 with the conception of the Kaufman mercury-propellant electron-bombardment thruster(17) at NASA/Lewis. Many of the design principles of the original Kaufman thruster are still used today, but there have been a number of new concepts developed which have led to cesium-propellant electron-bombardment microthrusters(18) having excellent performance.

Although some of the basic mechanisms important to the operation of electron-bombardment thrusters are not well understood, the basic technology of all components is well in hand. Electron-bombardment thrusters have been duration tested for many tens of thousands of hours in ground vacuum facilities. The most successful space flight test has been accomplished with the original Kaufman thruster. The SERT-II flight tests to be flown in the near future will have electron-bombardment thrusters in the propulsion system(2). There is no doubt that the electron-bombardment thruster stands today in the flight-prototype classification.

In the cesium electron-bombardment thruster, the cesium propellant flows by capillary action to the vaporizer and flow control. Cesium vapor then flows through the feed tube to the auto-cathode. The auto-cathode is equipped with a heater needed only in starting the thruster. The cesium covers the surface of the auto-cathode thereby enhancing the thermonic emission of electrons needed for the discharge in the ionization chamber proper. Electrons from the auto-cathode are attracted toward the anode, which is about 8 volts above cathode potential. The cylindrical-shell permanent magnet produces a solenoidal magnetic field which tends to keep the electrons in the ionization volume until they have had collisions. The ionization potential of cesium is about 3.5 ev, so the electrons have sufficient energy to produce a nearly fully ionized plasma in the chamber. The chamber and screen grid are maintained at a net accelerating voltage above space potential, and the accelerator electrode is maintained at a voltage below space potential. The electrostatic field between these electrodes causes a curved plasma sheath to form upstream of each of the holes in the screen grid. Ions are extracted from the plasma through this sheath and then accelerated out through the holes in the accelerator electrode to form ion exhaust beams. Electrons emitted from the neutralizer filament enter each of the ion exhaust beams to neutralize the positive space charge.

Thrust density of the electron-bombardment thruster is limited by the accelerator voltages and the accel lengths in the same manner as the contact-ion thruster. Flight-prototype electron-bombardment thrusters have accel lengths of several millimeters, so the net accelerating voltage must be in the kilovolt range in order to achieve a reasonably high thrust density. Power conditioning is required to convert the low voltage power from the solar-cell arrays to the high-voltage power required by the electron-bombardment thruster.

Flight-prototype electron-bombardment microthrusters(18) are shown in Figure 8. Chamber diameters of these thrusters are from 0.5 to 5.3 inches, and the thrusters cover a thrust range from 10 to 6700 micro-pounds.

Summary of flight-prototype electric thruster systems. Total system weights of the flight-prototype electric thruster systems are summarized in Figures 9 and 10; these figures are for propulsion times of 1000 to 10,000 hours respectively. A propulsion time of 10,000 hours corresponds to a mission time of somewhat over 1 year for the synchronous satellite illustrated in Figure 2. From inspection of Figure 9 it is evident that the resistojet is superior with respect to total system weight for short missions, but the electrostatic ion thruster systems become competitive when propulsion time reaches about 2000 hours. For a propulsion time of 10,000 hours the 500 micro-pound resistojet listed in Table I has a total system weight of 136 pounds, which is about 3 times as heavy as electrostatic ion thruster systems at that thrust level, as shown in Figure 10. On the basis of these total system weight comparisons it is evident that the resistojet electric propulsion system is superior for short missions, while the electrostatic ion thruster systems are superior for long auxiliary propulsion missions.

Other factors are also important in the choice of the electric propulsion system. The resistojet can operate directly from a solar-cell array power output, and therefore requires a minimum of power conditioning and controls. A typical power conditioning subsystem for the electrostatic ion thruster has more than 1000 critical components; such a large number of critical components is bound to drastically affect the reliability of the electrostatic ion microthruster system. In this respect the electrostatic ion thruster system can be competitive with the resistojet system only if adequate reliability of power conditioning is conclusively proven.

Another important factor for many missions is the capability for precision thrust vectoring. The contact-ion electrostatic thruster possesses this capability. Precision thrust vectoring might be possible for the electron-bombardment microthruster, but it is unlikely that precision thrust vectoring can be developed for the resistojet by any other means than mechanical gimballing.

The performance characteristics of the flight-prototype electric propulsion systems described here are firm, and form an excellent basis for judging the relative merits of the experimental thruster systems and the advanced concepts to be described in the following sections.

Experimental Thruster Systems

There are three thruster types judged to be in the experimental thruster category; charged-particle electrostatic thrusters (colloid thrusters), magnetoplasmadynamic thrusters (MPD thrusters), and solid-propellant electric thrusters (SPET systems). All of these experimental thrusters have received a considerable amount of research and development effort. Laboratory prototypes of each of these thrusters have been operated for hundreds of hours in ground vacuum facilities. Although the basic mechanisms important to the operation of each of the thruster types are not completely understood, performance measurements have been made with a sufficient degree of confidence to allow a fairly reliable evaluation.

Colloid thrusters. Colloid electrostatic thrusters have been advocated for some time as being potentially superior to any other type of electric thruster(19,20). In the past few years a liquid-spray charged-particle electrostatic thruster(21) has been developed to a status of excellent experimental performance.

In the laboratory prototype of this thruster, glycerol propellant is forced under pressure to flow into hollow needles with 0.014 inch outside diameter and 0.004 inch inside diameter. The glycerol propellant contains an additive such as sulfuric acid or sodium iodide to control the electrical conductivity of the fluid. When the liquid reaches the end of the tube, it is exposed to a strong electric field. The forces exerted on the liquid surface at the needle tip include surface tension, the feed pressure, and the electric pressure produced by the field. In a manner not yet well understood an unstable liquid is produced from which charged multimolecular droplets are ejected continuously. The charged droplets are then accelerated in the electrostatic field through the accelerator electrode aperture where electrons are added to the exhaust beam to neutralize the positive space charge.

Although the charged particles have a spectrum of charge/mass, this spectrum is fairly narrow so that the power loss is only about 10 to 30 percent due to the velocity distribution in the exhaust beam. The width of the charge/mass spectrum is somewhat independent of the specific impulse, and appears to be more a function of the geometry of the needle tips and of the uniformity of the needle bore. The mean value of the mass/charge can be controlled by adjusting the mass flow rate and the voltage between the needles and the extractor electrode, and by adjusting the electrical conductivity of the propellant and the type of additive used in the propellant. Charge/mass ratios of from 200 to 10,000 coulombs/kilogram have been obtained experimently; these values of charge/mass correspond to specific impulses of 140 to 1000 seconds respectively with a net accelerating voltage of 6000 volts.

There has been some difficulty with erosion of the needle tips, normally because of electric discharges. Even in the absence of needle tip erosion a gradual decrease in performance has been observed in some tests of 100 hours or more in duration. This gradual degradation in performance appears to be due to either deposits of material at the needle tip, or condensation of organic material on the extractor and the needle.

Estimated performance characteristics of the liquid-spray charged-particle electrostatic microthruster are shown in Figure 11, based on published data for 6-needle and 37-needle designs.

A large part of the estimated system weight at 345 micro-pounds of thrust is propellant weight, so if a propulsion time of only 1000 hours were required, the total system mass would be 12.4 pounds at this thrust level. This weight is less than that of flight-prototype resistojet system described previously, so it appears that the charged-particle electrostatic micro-thruster may become competitive with the resistojet system for short propulsion-time missions.

From inspection of Figure 11 it is evident that the charged-particle experimental microthruster will be competitive with flight-prototype ion thrusters if adequate durability of the experimental thruster can be demonstrated in the future. However, for missions with propulsion times of more than 10,000 hours, it is clear that the charged-particle electro-static thruster system weight will become higher than the flight-prototype ion thruster system weight, unless the specific impulse of the charged-particle thruster can be substantially increased by further research and development.

An important feature of the charged-particle electrostatic thruster is the low power consumption. By comparison of Figures 10 and 11 it is evident that the flight-prototype ion thrusters require much more power than the charged-particle electrostatic thruster, and this is true even if the specific impulse of the charged-particle electrostatic thruster is increased in the future. The fundamental reason for the higher power consumption of flight-prototype ion thrusters is simply that they are very inefficient. In the contact-ion thrusters large amounts of power are lost by radiation and conduction from the extremely hot ionizer. In the electron-bombardment thrusters it appears that a great amount of power is lost by recombination of ions on the inner walls of the thruster chamber. In the charged-particle electrostatic thruster the only appreciable power loss is in the vaporizer, the neutralizer, and the exhaust velocity distribution. If the charged-particle electrostatic thruster can be developed to a flight-prototype status then it might be possible to supply the moderate power requirements of this thruster with radioisotope thermoelectric generaters. For vehicles and missions where solar cells are awkward or impossible the low power consumption of the charged-particle electrostatic thruster could be a significant advantage.

Solid propellant electric thrusters (SPET). The SPET microthruster(22) is an unusual concept in which a thin film of propellant is exploded by the passage of a sudden discharge of electrical energy. The exhaust velocity of the resulting plasma can be augmented by electromagnetic acceleration.

In the SPET thruster, capillary flow carries the propellant to a slot connecting two electrodes. When a capacitor containing electrical energy of about 1 joule is discharged across the two electrodes the sudden addition of electrical energy explodes and ionizes the propellant which can be further accelerated by the parallel rail arrangement. The normal mode of operation is one pulse every two seconds.

One of the difficulties encountered in the development of SPET has been the formation of hard deposits of certain constituents of the organic propellants. It is reported that a non-organic propellant has been found which has appropriate characteristics for explosion by electrical energy addition. Duration tests have been conducted with as many as 3.65 million pulses. There are 31.5 million seconds per year, so these duration tests are equivalent to about 0.1 year at the rate of 1 pulse per second. Data(22) from this duration test are listed in Table II.

If the SPET experimental microthruster can be developed further to have a durability of one year then it could provide about one tenth of the total impulse required for the attitude control of the 2000-pound synchronous satellite illustrated in Figure 2. If still further development is possible, and the SPET system is scaled up to the equivalent of 14 micro-pounds of thrust, then the input power to the thruster would be 10 watts. This would bring the SPET microthruster to a competitive status with the flight-prototype contact-ion microthruster shown in Figures 9 and 10. Scaling up to this thrust level involves the release of about 10 joules of electrical energy per pulse; whether the release of this much energy into the propellant film can be accomplished without damage to the thruster is a question that still must be answered.

Another difficulty encountered in the development of the SPET microthruster is the lifetime of the switching tube. Cold cathode thyratrons have been used so far, but these have a life of only about 200,000 pulses, which is a far from adequate durability.

On the basis of this information it must be concluded that a considerable amount of development remains to be done before the SPET microthruster can achieve a flight-prototype status.

MPD thrusters. Electromagnetic acceleration of plasmas has been the subject of much research since the early days of electric propulsion. Two rationales have often been quoted for this research effort, one being that the plasma thruster has promise of a high thrust per unit area compared with the ion thrusters, and the other being that the plasma thruster can operate on low-voltage power. High thrust per unit area is not a particularly important advantage for microthrusters, but operation with low-voltage power may be a significant advantage.

For a number of years a magnetoplasmadynamic (MPD) thruster has been under development at the NASA/Lewis Research Center(23,24). In this MPD thruster, a propellant such as argon flows into a hollow cathode where it is ionized by electron bombardment. The major portion of the propellant enters the chambers through the anode feed tubes and is ionized in the discharge that exists between the cathode and the anode. A strong axial magnetic field is maintained by the permanent magnets and the electromagnets. Field strength at the cathode is about 500 gauss and is about 250 gauss at the exhaust aperature. Thrust is generated by the expansion of the hot plasma in the magnetic nozzle. The discharge voltage ranges from 100 to 200 volts, depending on the operating conditions.

In earlier versions of this MPD thruster the magnetic field was produced by electromagnets entirely. Because of poor heat rejection the electromagnets overheated after about 15 minutes of operation; electromagnet power was 300 watts at a coil temperature of 600 deg. C. In the EM-PM MPD thruster a cylindrical permanent magnet around the hollow cathode generates a significant fraction of the total magnetic field. With this combination of permanent magnet and electromagnets, the electromagnet power has been reduced by 50%, but as a consequence the thruster weight has been increased significantly by the addition of the permanent magnet.

Estimated total system weights of the experimental MPD microthruster are compared in Figure 12 with weights of flight-prototype ion thruster systems. For missions with propulsion time of 10,000 hours the experimental MPD microthruster system would be 2 or 3 times heavier then the flight-prototype ion thruster systems. Even if the electromagnet power were eliminated entirely the MPD microthruster system would not be competitive. The same remarks can be made for the comparative status of the experimental MPD microthruster in missions with propulsion times of 1000 hours. On these grounds, it must be concluded that the EM-PM MPD thruster does not have much promise of having lower system weights then the flight-prototype ion thruster systems.

Improvement of performance of this thruster would probably have to be accomplished with propellants other than argon. The estimated system weights do not include weight penalties associated with the use of gaseous propellants. From inspection of Figure 12 it is clear that a tankage of 50 to 100% would seriously increase the total system weight of the MPD microthruster for missions with propulsion times of the order of 10,000 hours. In principle, propellants such as the alkali metals could be used in the MPD microthruster, and these propellants would have much lower tank weights.

A potential advantage of the MPD microthruster is the elimination of power conditioning with resulting reliability due to simplicity. Realization of this advantage must await the development of flight-qualified solar-cell arrays(8) that can operate at output voltages of 100 to 200 volts. Until such solar power is available, the EM-PM MPD thruster cannot be considered a serious competitor, and even then the MPD thruster performance would have to be improved significantly.

Advanced Concepts

There are two advanced concepts in auxiliary propulsion that appear to have great promise of much better performance than the flight-prototype ion thruster electric propulsion systems. These are the radioisotope-heated contact-ion microthruster, and the plasma-separator microthruster. Both of these advanced concepts are based on demonstrated principles, but have not yet been operated as laboratory prototype thrusters. In fact, the feasibility of all components of these advanced concepts has been demonstrated in laboratory experiments. There are a number of other advanced concepts such as the ELM (exploded liquid metal) thruster, the REPPAC (pulsed plasma accelerator) thruster, pulsed vacuum-arc microthruster, etc. None of these are included in the present review because of the lack of sufficient information to establish their feasibility or potentiality.

Radioisotope-heated contact-ion microthrusters. A major fraction of the power requirements for contact-ion thrusters is consumed by the ionizer heater. The porous-tungsten ionizer(25) must be maintained at 1150 deg. K for an ion current density of 1 milliampere/sq. cm. and 1350 deg. K for an ion current density of 10 milliampere/sq. cm. Because of radiation and conduction heat transfer a considerable amount of power must be used to maintain the ionizers at these temperatures. It appears possible, in principle at least, to use radioisotope heating of ionizers instead of the electrical heating that is presently used in flight-prototype contact-ion microthrusters.

Power requirements for the various components of the flight-prototype contact-ion microthrusters are listed in Table III. For the 14 micro-pound microthruster, the ionizer heater requires 7.6 watts which corresponds to a solar cell power of 11.7 watts which in turn corresponds to about 1.8 lb. of solar-cell array. From this it is clear that radioisotope heating of 10 micro-pound thrusters will save only a small amount of total system weight. For the 300 micro-pound micro-thruster, about 7 lb. of solar-cell array could be saved. From inspection of Figures 9 and 10 it is evident that a 7 lb. reduction in total system weight is appreciable.

On the basis of this cursory examination of the merits of radioisotope heating of ionizers it may be concluded that significant weight reductions could be attained if the radio-isotope containment and helium-gas-receiver vessels can be developed to a reliable and lightweight status. It is expected that the technology being developed for radioisojets(26) will have much bearing and usefulness in radioisotope heating of ionizers. The promethium-147 heater capsule in the radioisojet has already attained operating temperatures of 1280 deg. K. In the static heat transfer environment of ionizer heating, and with improved thermal insulation, it is possible that promethium-147 could become an adequate heater for contact-ion microthrusters at low ion current densities. Since promethium-147 is a beta particle emitter and has only a small radiation intensity, it has operational advantages over the alpha particle emitting radioisotopes. Further theoretical analysis and experimental development is needed before the feasibility of radioisotope heating of ionizers can be established conclusively. Additional work will be worthwhile because of the potential reductions in system weight in the several hundred micro-pound thrust range.

Plasma separator thruster. Some experimental work has been done which establishes the performance of the critical components of the plasma-separator thruster concept. Although this concept has not been operated as a complete thruster the components appear to be fully compatable, and their measured characteristics offer promise of a high performance thruster(27).

In the plasma-separaor microthruster concept, propellant is vaporized and flows through a hollow cathode. The surfaces of the hollow cathode are very hot to insure copious thermionic emission of electrons, and this emission is further enhanced if the propellant is an alkali metal, which lowers the work function of the cathode surface. Electrons are attracted to the anode which is at a potential at about 10 volts above the cathode. Nearly complete ionization of the propellant occurs in the cathode and nozzle region; this ionization is accomplished with a small amount of power because of the high density of the propellant vapor in the hollow cathode. (In contrast, ionization in the Kaufman electron-bombardment thruster occurs at a much lower density). The nearly fully ionized plasma formed in the hollow cathode expands rapidly though the nozzle and the expansion region; this expansion occurs quickly so there is negligible recombination. The portion of the total ion flow reaching the separator electrode is extracted from the plasma and accelerated electrostatically to the desired exhaust velocity. The peripheral flow of ions is condensed on the cone and returned to the feed system for another pass through the hollow cathode. By virtue of special design and operating conditions of the accelerator electrodes, all of the ions approaching the separator electrode are extracted and accelerated into the exhaust beam.

Performance characteristics of the hollow-cathode plasma source have been established with extensive experimental measurements(27). The electric power consumption of the hollow-cathode plasma source is about one-tenth of the electric power consumption in the Kaufman electron-bombardment ionization chamber. (As reported in another paper(28) a hollow cathode is being used in the SERT-II thruster and has resulted in considerable improvement in performance). The total-throughput ion optics of the accelerator system has received sufficient experimental investigation to verify the concept. On the basis of these facts it is concluded here that the plasma-separator concept is a valid and realistic thruster design.

The possibility of operating the plasma separator thruster directly from the output of the solar-cell arrays is being investigated(29). The standard output voltage from solar-cell arrays is 28 volts d.c. When mercury or cesium are accelerated through a potential drop of 28 volts, their final velocities are very low, which results in very low specific impulses. The alkali metals potassium, sodium, and lithium are much lighter than cesium and would produce satisfactorily high specific impulse in a 28-volt accelerator. Since propellants such as these would also lower the work function of the hollow cathode surface, it is reasonable to expect that the high performance of the hollow-cathode plasma source would be preserved.

With a low accelerator voltage, the thrust density will be small unless the accel length can be made very small. For acceleration of lithium, sodium, and potassium through 28 volts, the thrust density is reasonably high if the accel length is about 0.010 inch. Accel lengths of less than 0.040 inch were considered to be impractical until some excellent work in West Germany on accelerator structures was reported(30) in early 1966. This work showed that a solid dielectric insulator could be sandwiched between the separator electrode and the accelerator electrode. Electrode systems such as this have been tested(31) on a mercury electron-bombardment thruster and have been found to be satisfactory. These tests were of comparatively short duration, and it has not been established whether adequate durability could be obtained using mercury propellant and several hundred volts in the accelerator (in order to obtain a reasonable value of specific impulse with mercury).

Since the lighter alkali metals are being considered for propellants in the plasma-separator microthruster, the accelerator voltage can be much less than for mercury propellant, perhaps as low as 28 volts d.c. The threshold for sputtering in most ion/target-metal combinations is about 20 e.v.; and since most of the charge-exchange ions will have energies considerably less than the total voltage in the accelerator, then it may be concluded that a 28-volt accelerator system will have essentially no sputtering problem. With the problem of sputtering being eliminated by low-voltage operation, there should be no problem with accelerator electrode durability.

Estimates have been made of the potential performance of the plasma-separator microthruster concept; these estimates(29) are based on the experimental performance of the hollow-cathode plasma source. Experimental data for the hollow-cathode plasma source have all been obtained with cesium propellant. It is assumed in the estimated performance calculations that the power consumption in the hollow-cathode plasma source is proportional to the first ionization potential of the propellant, so that the plasma source performance is lower for the other alkali metals than it is for cesium. It is also assumed in the estimate that only 60% of the plasma issuing from the hollow cathode arrives at the separator electrode, while the other 40% must be recirculated. With these conservative assumptions, the estimated performance of this microthruster system is shown in Figures 13 and 14.

It is important to note that the estimated performance in Figures 13 and 14 is for an accelerator voltage of only 28 volts d.c. If 100 volts d.c. were available from the solar-cell array, then the specific impulse with each of the propellants shown in the figures would be doubled, thereby reducing the propellant requirements by 50%. In addition the power requirements would be reduced at each thrust level, thereby reducing the weight of the solar-cell array. Estimated total system weights with 100-volt solar-cell arrays is shown in Figure 15.

Even with only 28 volts d.c. accelerating voltage, the hypothetical plasma-separator microthruster system appears to have promise of much better performance than the flight-prototype ion microthruster systems. If the output voltage of the solar-cell arrays can be raised to 100 volts, then the hypothetical plasma-separator thruster has promise of much superior performance. It should be pointed out again that operation of microthrusters directly from the solar-cell array output would eliminate the need for the very complex power conditioning with its more than 1000 critical components. The potential advantages of this microthruster concept should be just cause for further development work.

CONCLUDING REMARKS

Electric propulsion is ready to take a useful part in a number of space missions. With solar-cell powerplants, electric thrusters of several types offer significant advantages over all other forms of propulsion for satellite station-keeping and attitude control. There are a number of experimental thrusters and advanced concepts that have promise of even better performance, so it may be expected that electric propulsion will become even more useful in the future. Present flight-prototype electric propulsion systems are rather complex, being composed of thousands of critical components in the power conditioning subsystem. Simplicity usually is synonymous with reliability, and this implies that the advanced concepts that can be directly coupled to the powerplant are of the most interest for continued research and development. With a firm flight-ready status, and with definite potential for great improvement, the future of auxiliary electric propulsion systems is sure to grow in importance and scope.

Not so long ago electric propulsion suffered outrageous jibes about long extension cords. Today electric propulsion stands ready to perform tasks in satellite missions that cannot be done by less sophisticated propulsion systems. The not too distant future may see the "heavens fill with commerce"(32) carried on with electric-propulsion satellites.

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Table I. - Performance characteristics of AVCO flight-prototype resistojet microthrust propulsion system

thrust, micro-pounds	500
propellant	ammonia
tankage	25%
specific impulse, sec	175
power watts	7

propulsion time, hrs.	1,000	10,000
thruster system, lb.	4.8	28.6
controls, lb.	.5	.5
propellant, lb.	10.6	106.
solar-cell array, lb.	1.1	1.1
TOTAL SYSTEM WEIGHT, lb.	17.0	136.2

Table II. - Performance data for the SPET experimental microthrust propulsion system.

number of pulses in duration test	3.65×10^6
equivalent operation time at 1 pulse/sec	0.1 year
total impulse per pulse	1.4 micro-pound-sec
equivalent specific impulse	3300 sec
efficiency	10%
total impulse per year at 1 pulse/sec	44 lb-sec/yr
equivalent thrust level	1.4 micro-pound
power input to thruster	1 watt

Table III. - Power requirements for contact-ion microthrusters.

thruster type	EOS	Hughes
thrust, micro-pounds	14	300
ion beam, watts	1.48	29.8
electrode drain, watts	.08	.2
vaporizer, watts	2.0	3
neutralizer, watts	1.1	6
sub total	4.66	39
ionizer heater, watts	7.66	30
total power to thruster	12.3	69

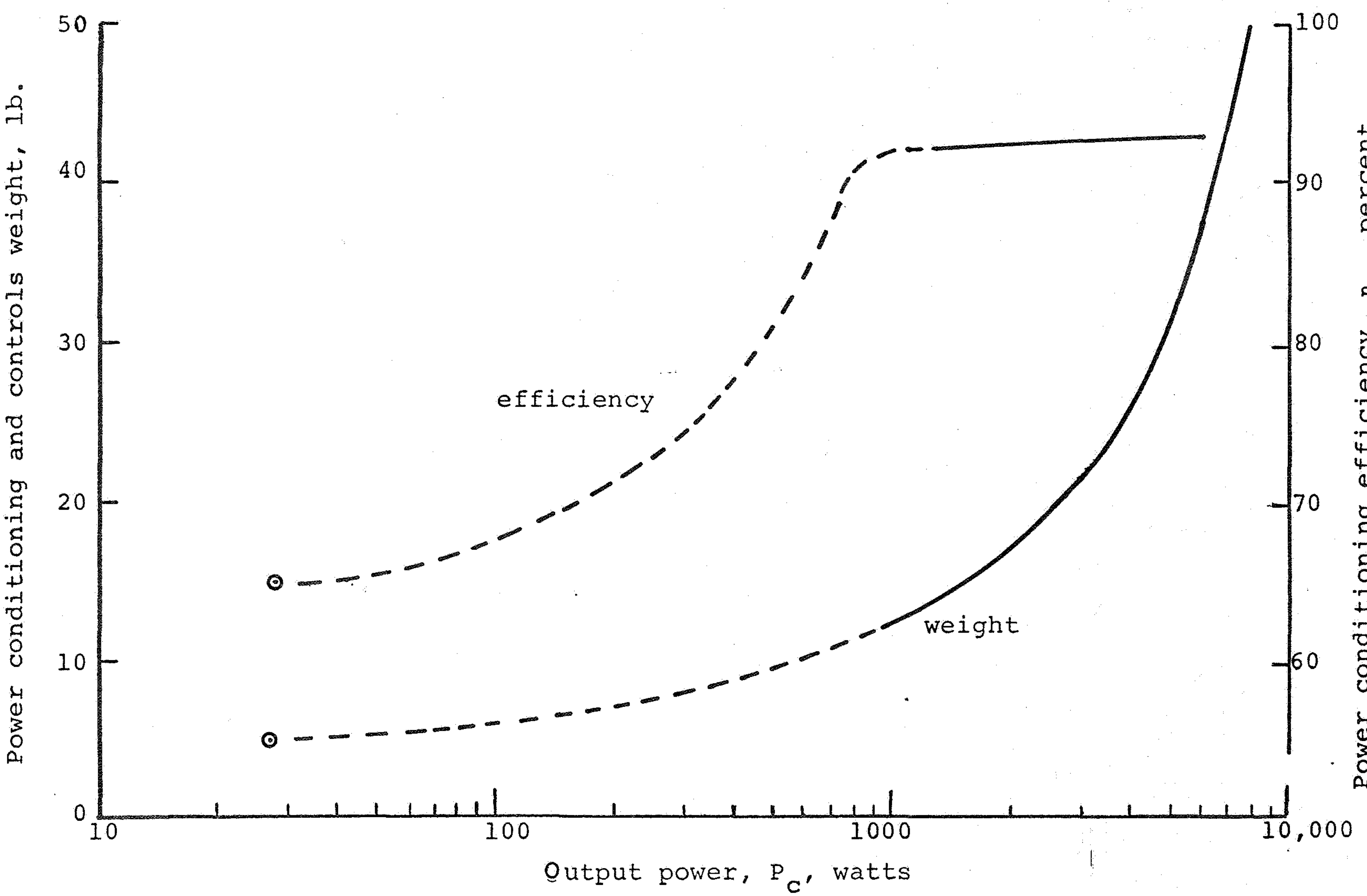

FIG. 1 - Power conditioning and control system weight and efficiency.

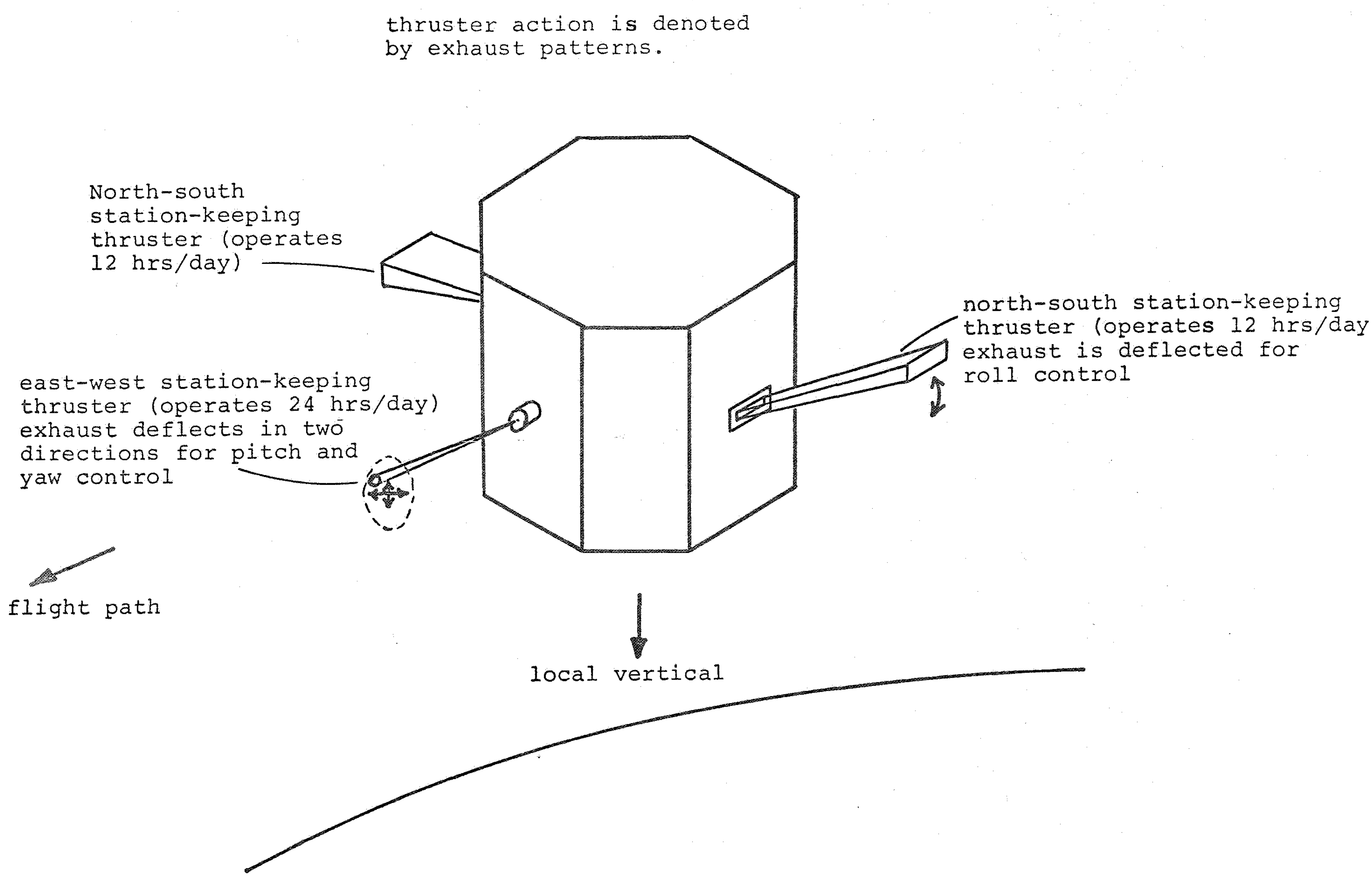

FIG. 2 - Electric propulsion system for station-keeping and attitude-control of synchronous satellites.

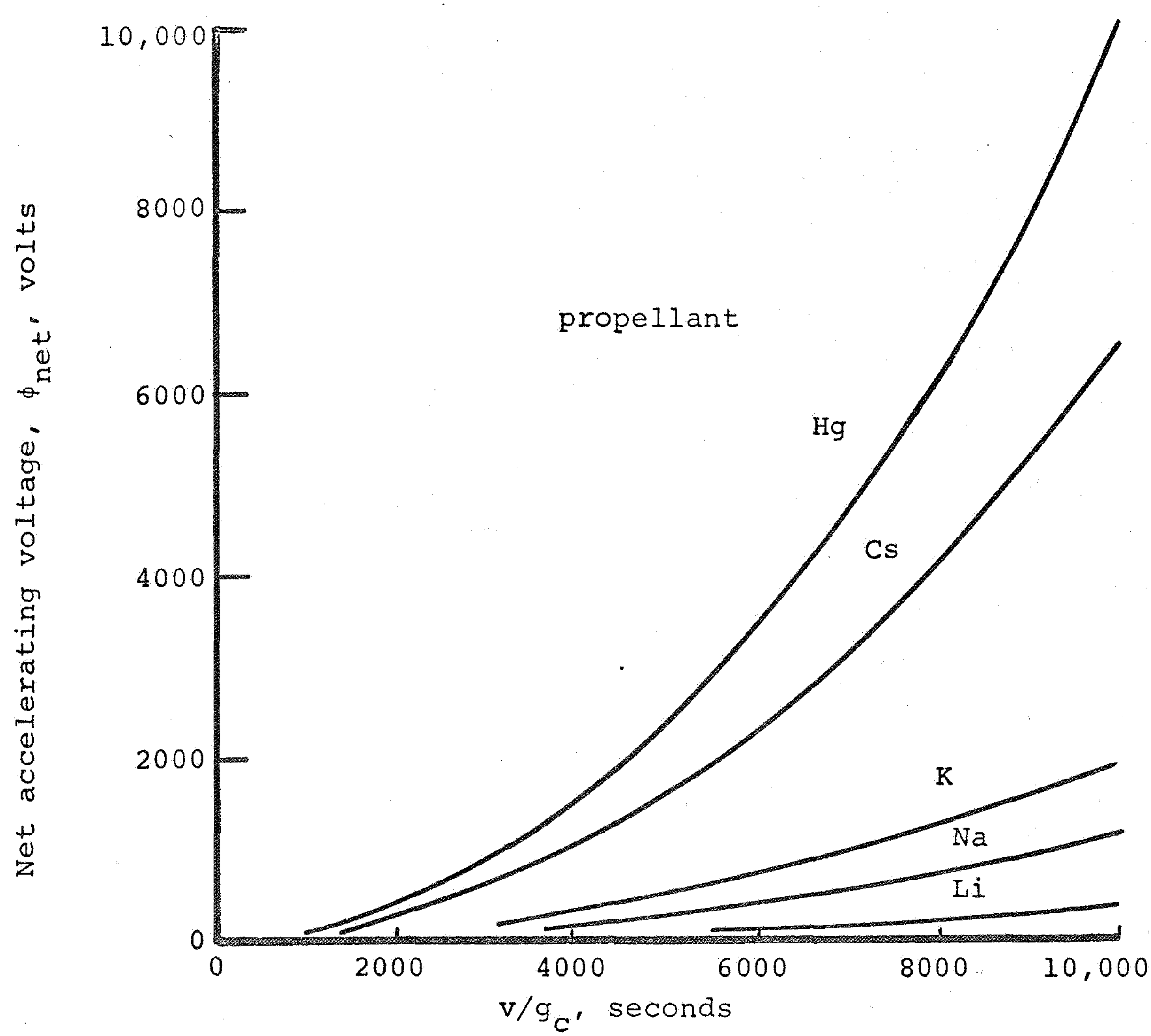

FIG. 3 - Net accelerating voltage for various singly-charged atomic-ion propellants. Specific impulse $I = \eta_U v/g_c$, where η_U is propellant utilization efficiency, and v is exhaust velocity.

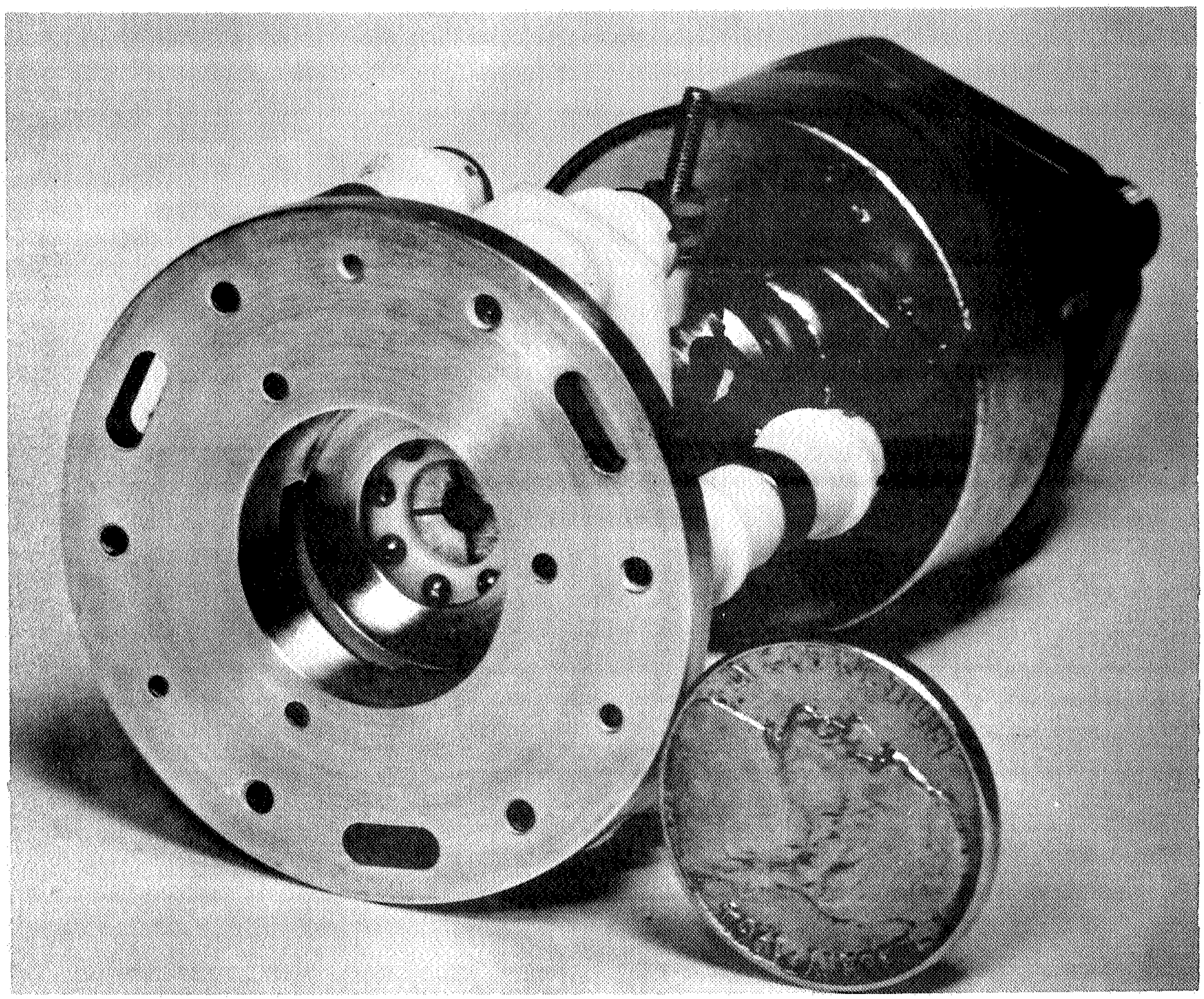

FIG. 4 Segmented electrodes for precision exhaust-beam vectoring in flight-prototype contact-ion microthruster. Courtesy of Hughes Research Laboratories, Malibu, California.

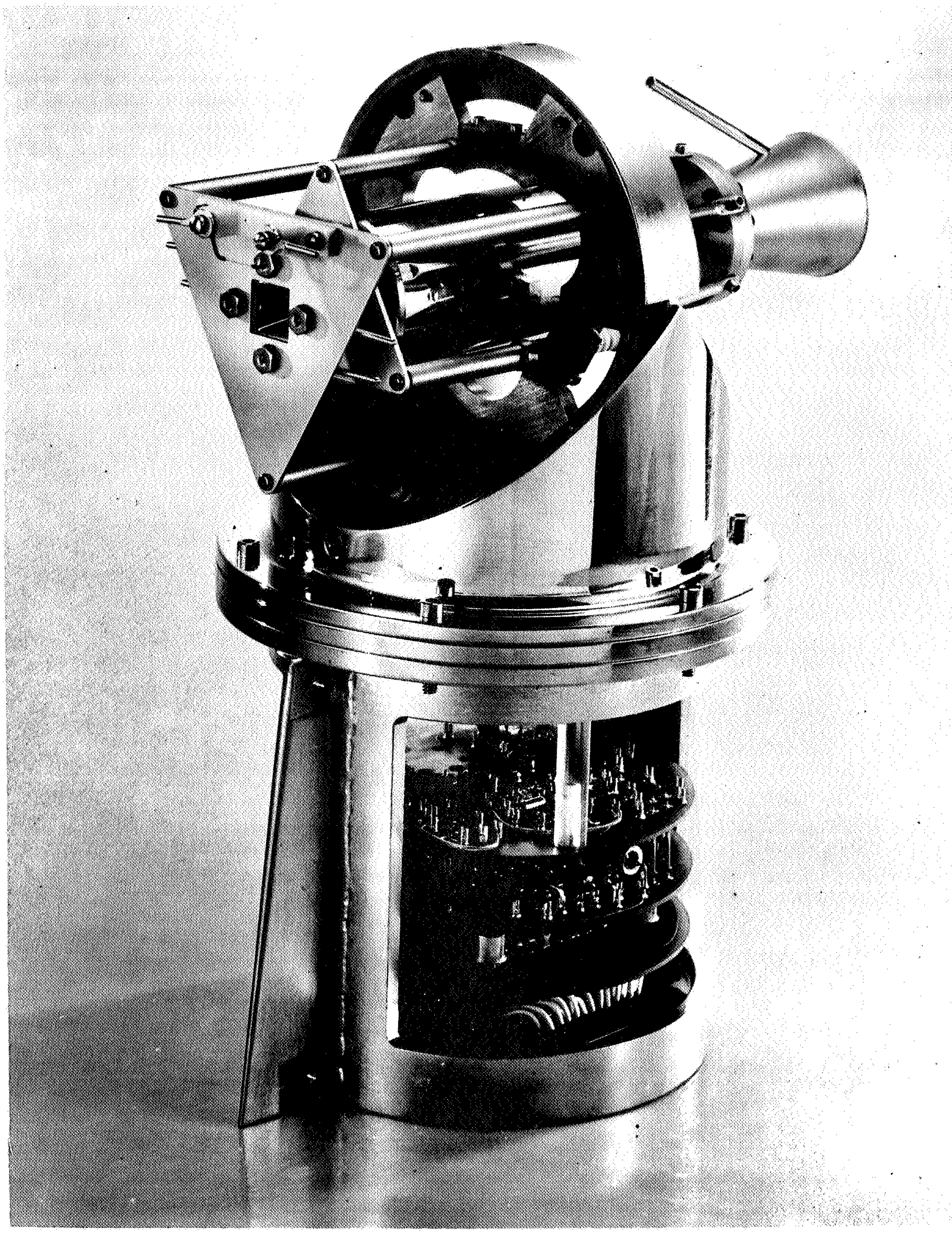

FIG. 5 Flight-prototype contact-ion microthruster system. Thrust: 14 micro-pounds. Courtesy of Electro-Optical Systems, Inc., Pasadena, California.

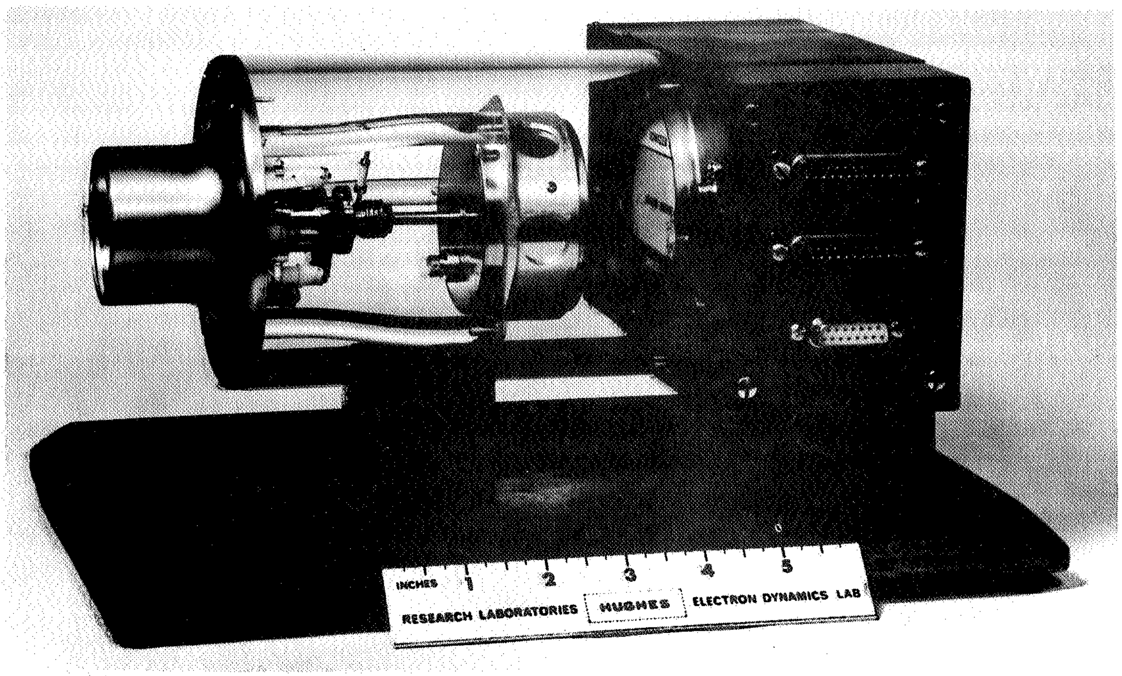

FIG. 6 Flight-prototype contact-ion microthruster system. Thrust: 20 micro-pounds. Courtesy of Hughes Research Laboratories, Malibu, California.

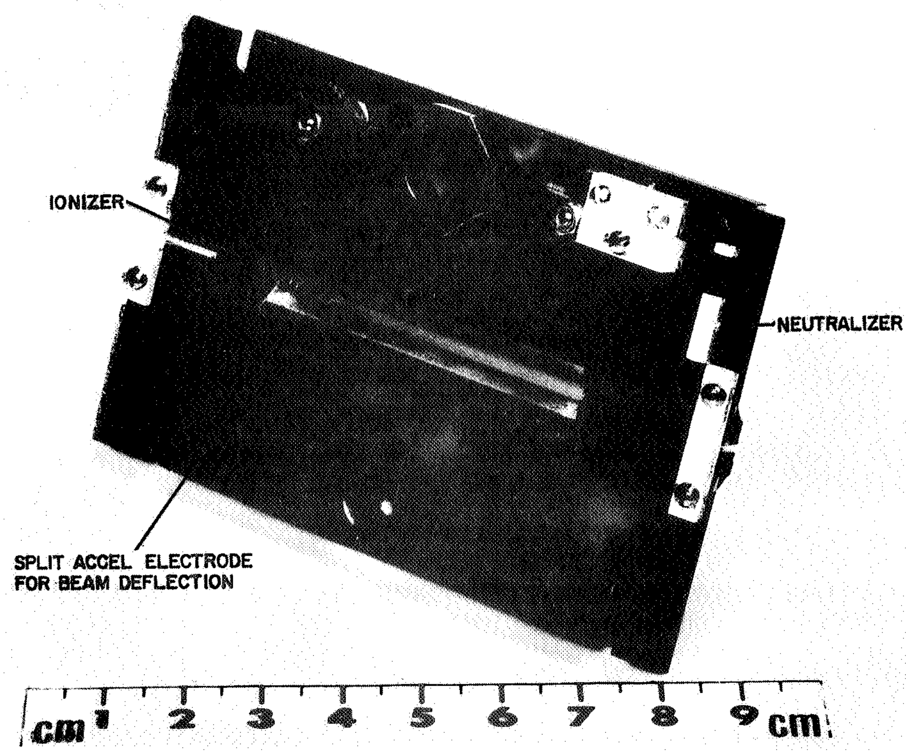

FIG. 7 Flight-prototype contact-ion microthruster. Thrust: 300 micro-pounds. Courtesy of Hughes Research Laboratories, Malibu, California.

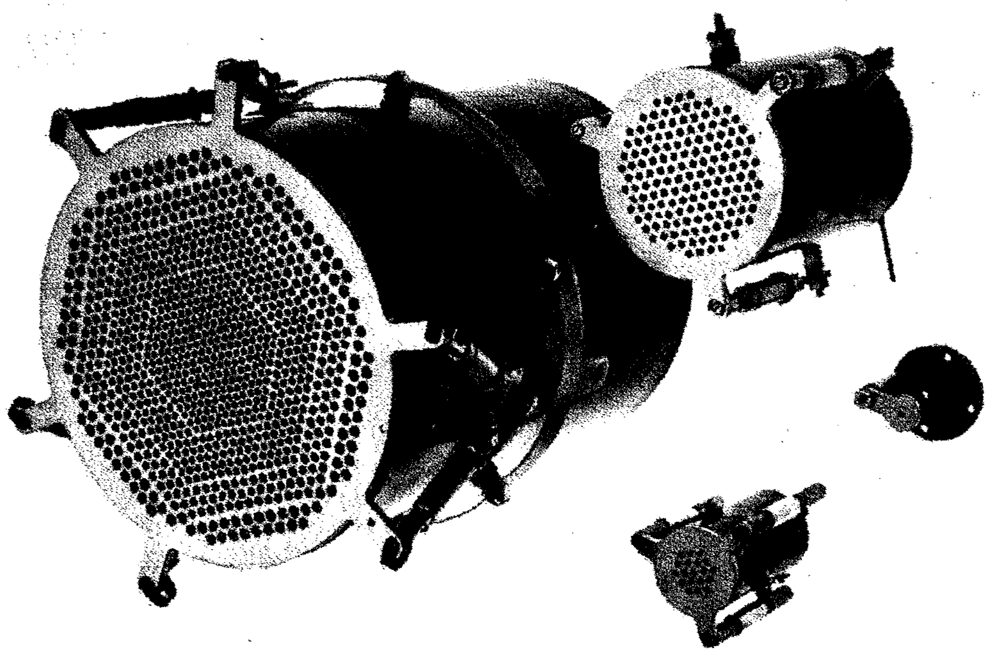

FIG. 8 Flight-prototype electron-bombardment microthrusters. Courtesy of Electro-Optical Systems, Inc., Pasadena, California.

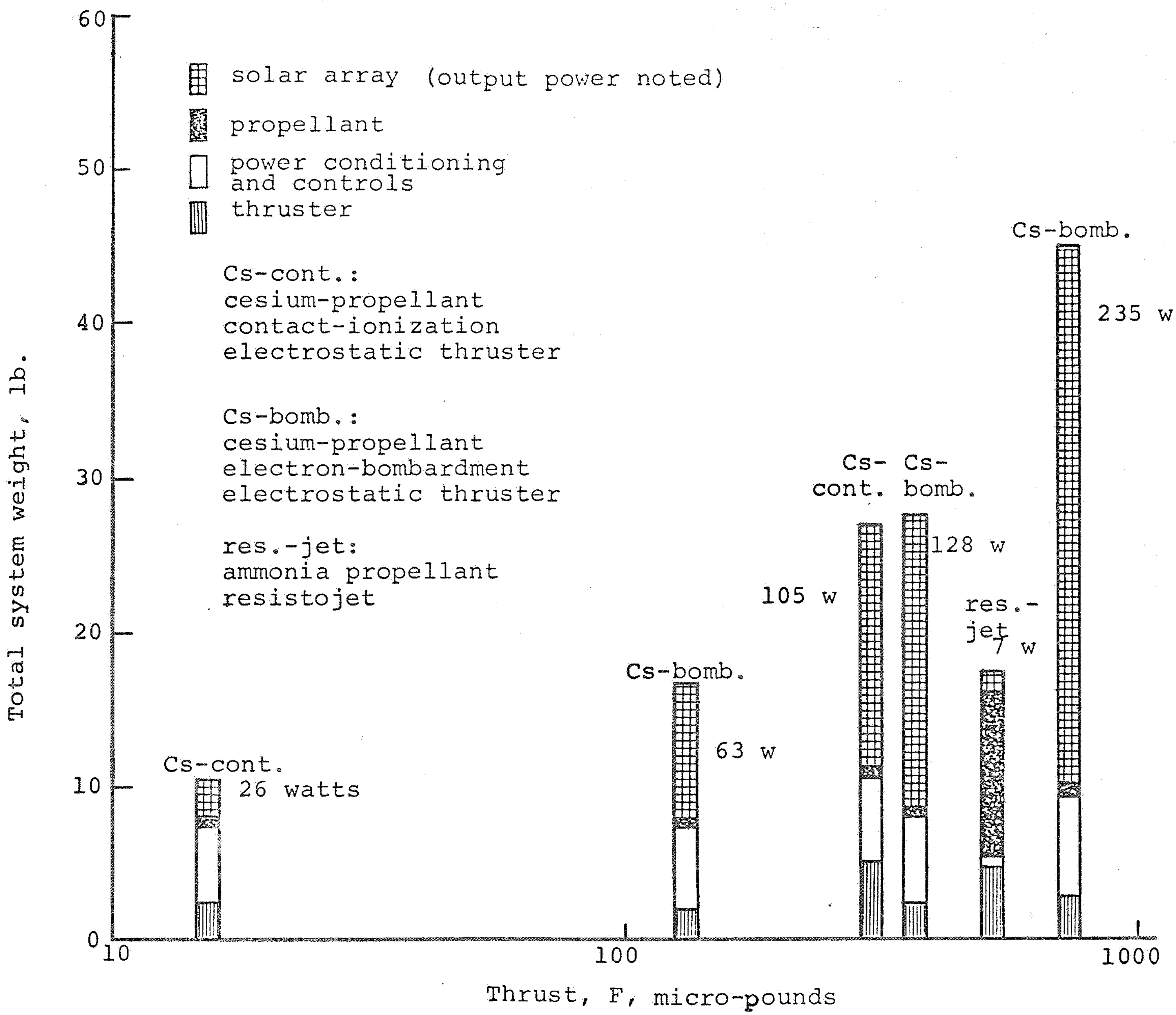

FIG. 9 - Flight-prototype ion thruster systems. Propulsion time: 1,000 hr.

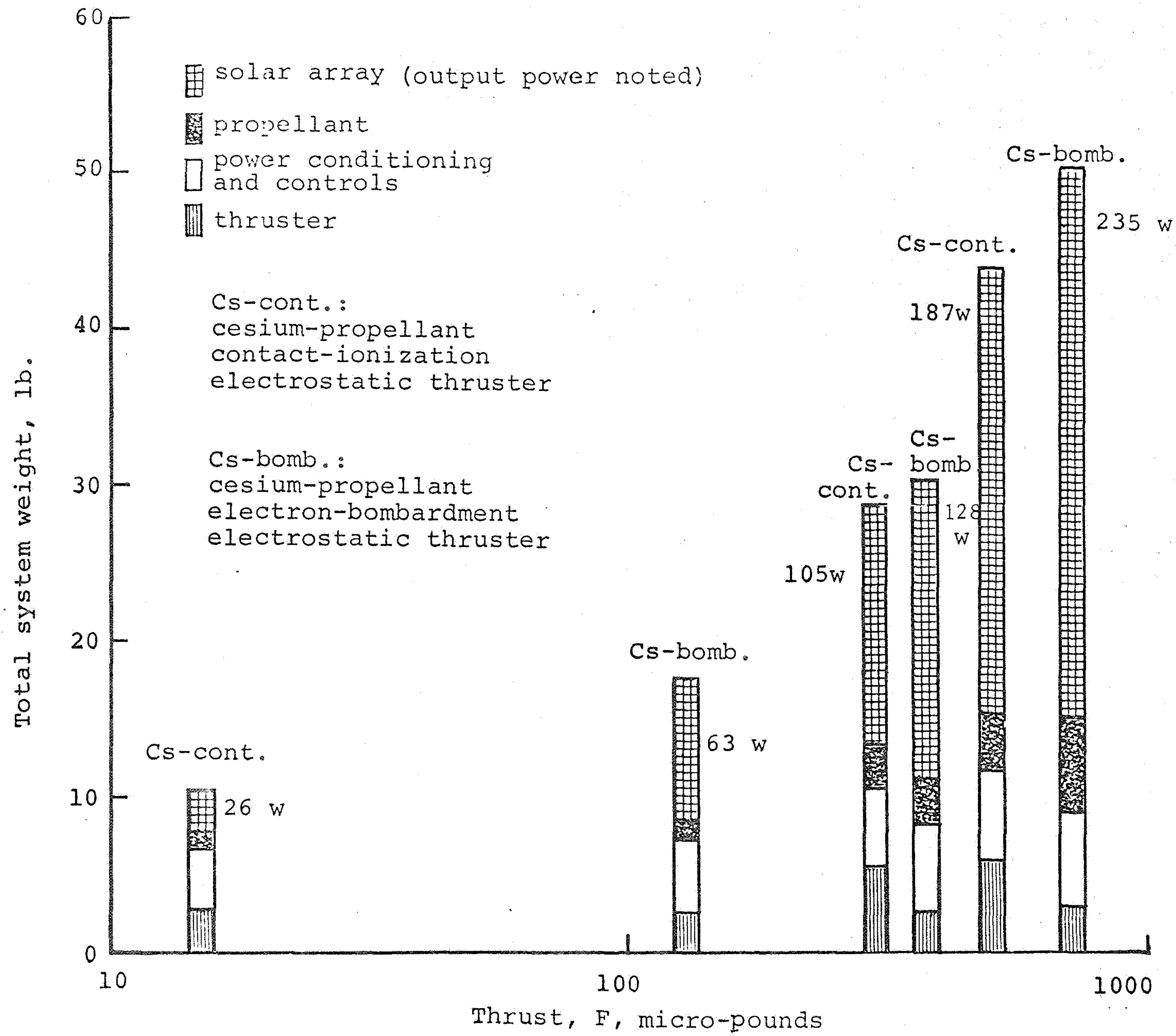

FIG. 10 - Flight-prototype ion thruster systems. Propulsion time: 10,000 hr.

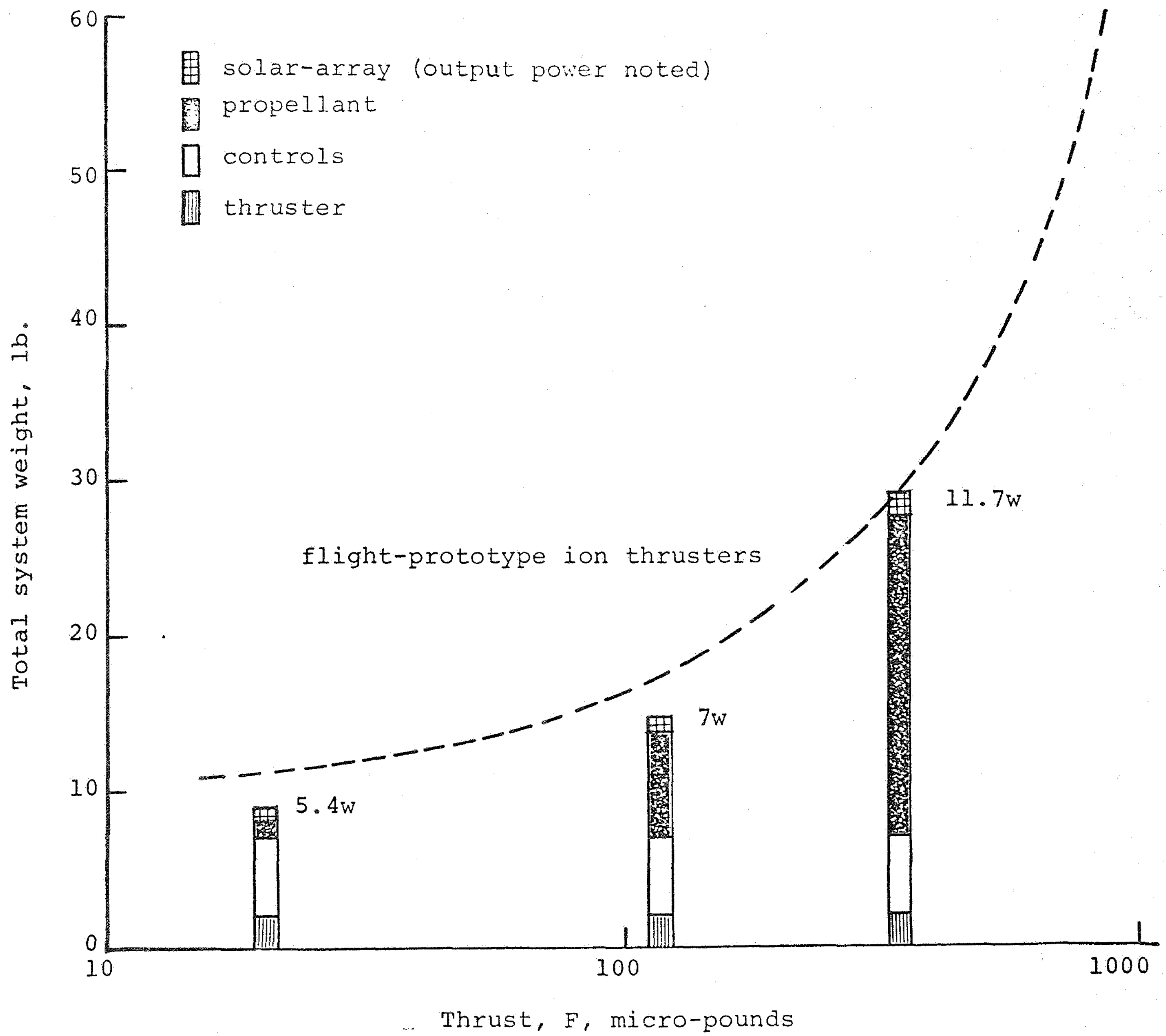

FIG. 11 - Estimated performance of experimental charged-particle electrostatic thruster in auxiliary propulsion systems. Propulsion time: 10,000 hrs.

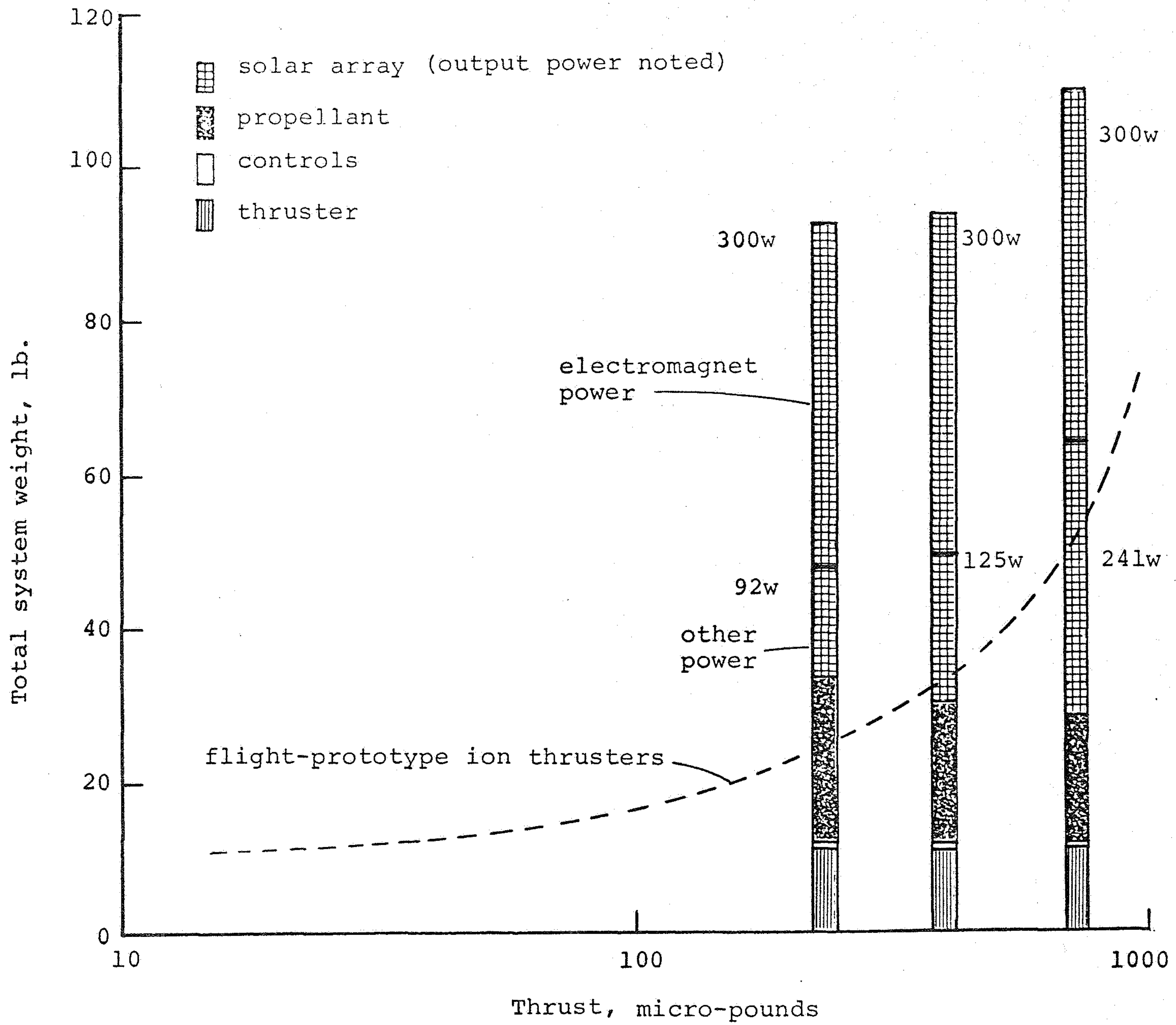

FIG. 12 - Experimental MPD microthruster. NASA-Lewis design, two-coil electromagnet and one permanent magnet. Argon propellant. Propulsion time: 10,000 hr.

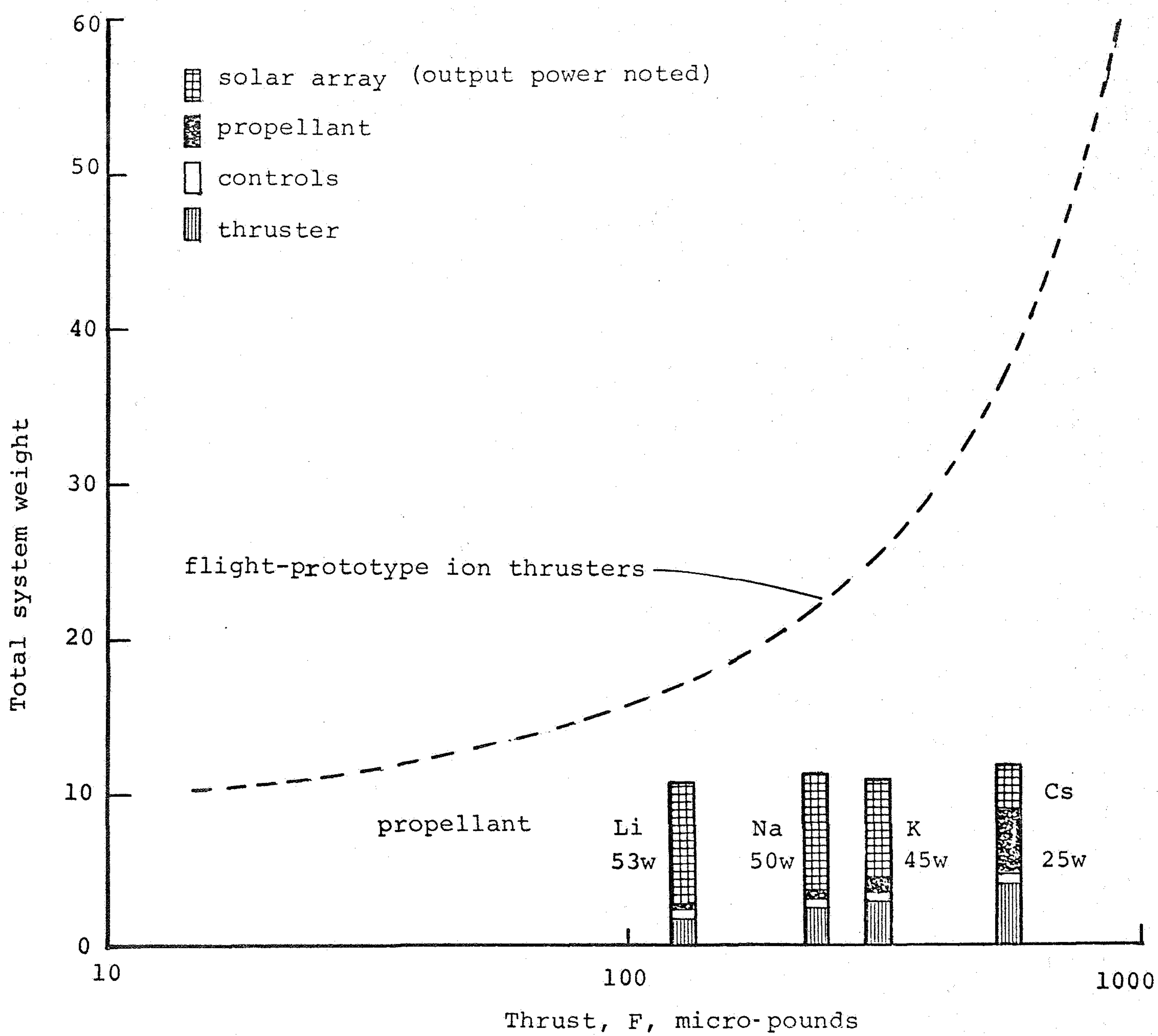

FIG. 13 - Hypothetical plasma-separator thruster systems. Accelerator voltage: 28 volts d.c Propulsion time: 1,000 hr.

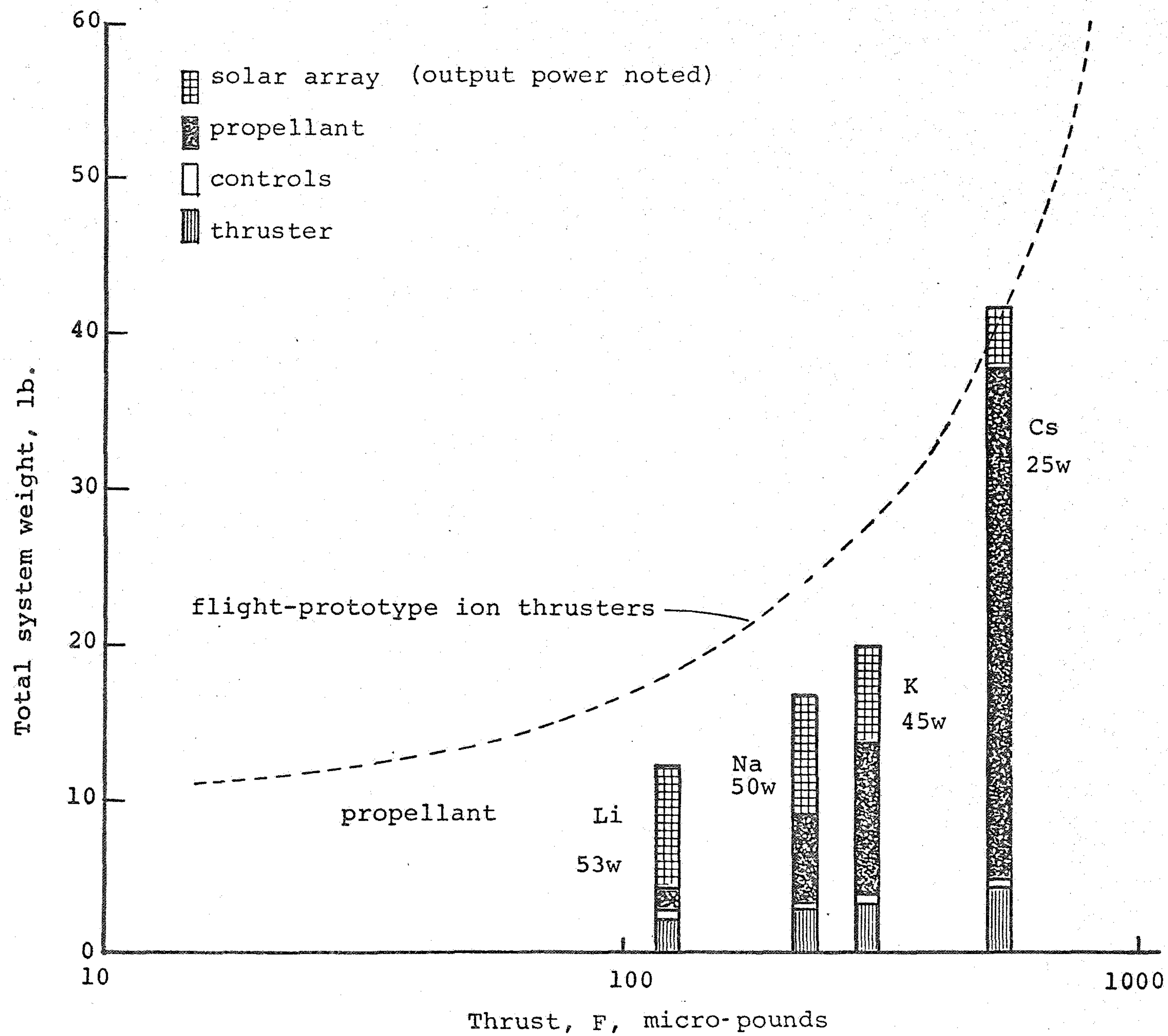

FIG. 14 - Hypothetical plasma-separator thruster systems. Accelerator voltage: 28 volts d.c. Propulsion time: 10,000 hr.

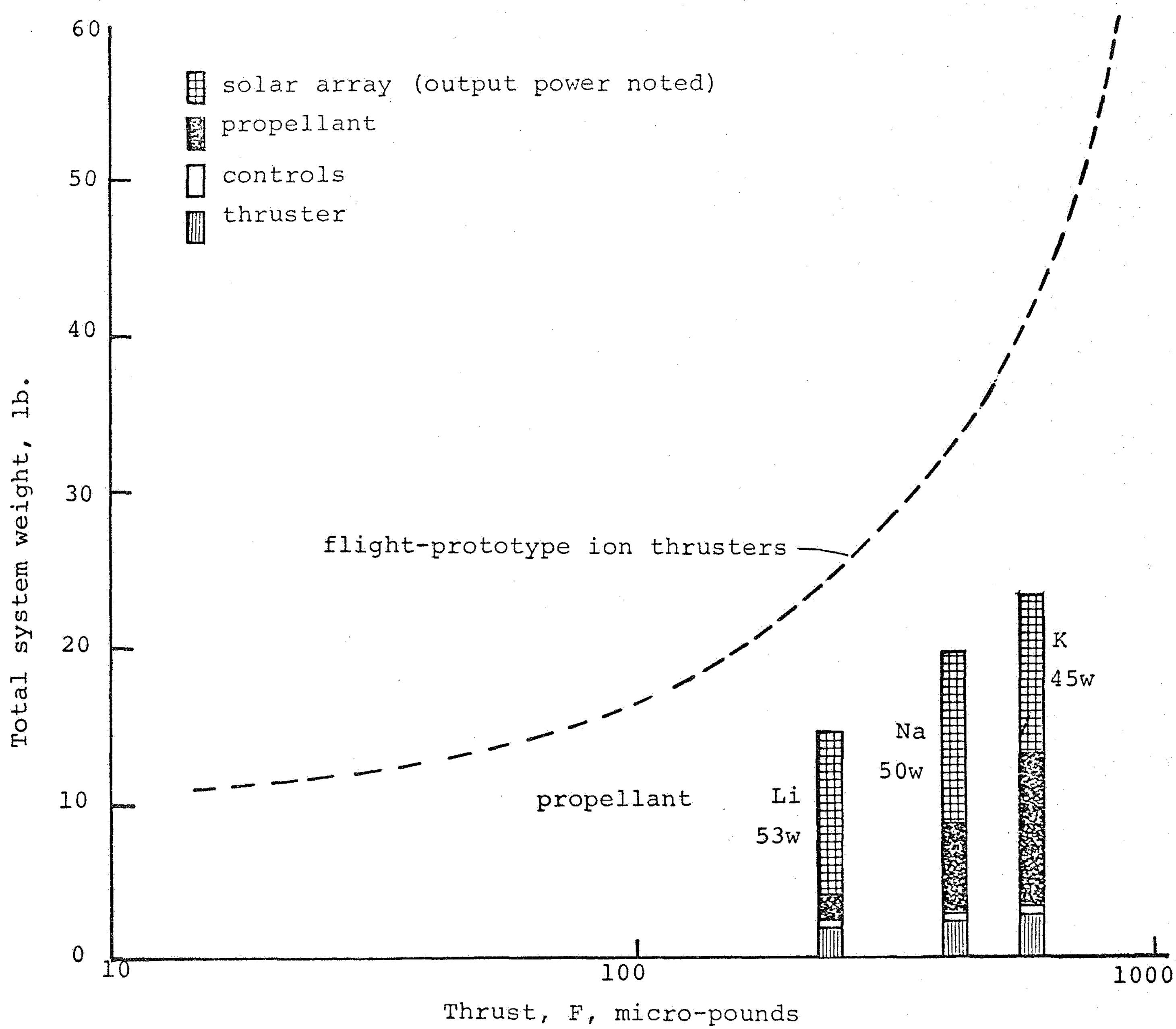

FIG. 15 - Hypothetical plasma-separator thruster systems. Accelerator voltage: 100 volts d.c. Propulsion time: 10,000 hr.